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Wings - Lift (99169); Lift - Mach number effect (54694); Wings, Swept-back - Lift (99306.4); Wings - Stal'ing characteristics (99179)

Aero-1867

Effects of Mach number on maximum lift

Royal Aircraft Establishment, Farnborough, Hants

Gt.Brit. Eng. Restr. Restr. Jan'47 27 table, graphs

Mach number effect on maximum lift is determined for unswept and swept-back wings. Swept-back wings show the same early tip stalling tendencies at high speeds as they do at low speeds. For unswept wings the $C_{L\,max}$ of sections with far back positions of maximum thickness is higher than that of conventional sections, because of the further back position of upper surface shock wave. At high Mach numbers thin airfoils have higher $C_{L\,max}$ than thicker ones.

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## EFFECTS OF MACH NUMBER ON MAXIMUM LIFT

by

F. N. KIRK, M.P.C., I.D.N.

RESTRICTED

Technical Note No. Aero.1867

January, 1947

ROYAL AIRCRAFT ESTABLISHMENT, FARNBOROUGH

Effects of Mach Number on maximum lift

by

F.N. Kirk, M.P.C., I.D.N.


## SUMMARY

### 1. Unswept Wings

The data on the effect of Mach number on $C_{L\ max}$ are scanty and their ad-hoc nature permits only two conclusions.

(a) At a given Mach number, the $C_{L\ max}$ of sections with far back position of the maximum thickness is higher than that of conventional sections, owing to the further back position of the upper surface shock wave.

(b) At high Mach numbers thin aerofoils have higher $C_{L\ max}$ than thicker ones.

Systematic research is needed and as the effect of Reynolds number on $C_{L\ max}$ appears to be small at high Mach number the systematic tests could be made say in the N.P.L. high speed tunnel. The most important parameters on which evidence of their effects is required are thickness-chord ratio, position of maximum thickness and camber.

No simple means of improving $C_{L\ max}$ at high Mach numbers can be suggested. Distributed suction over a portion of the upper surface may improve $C_{L\ max}$ as long as the suction is applied just behind the shock wave. The testing of such a device would be of value although its practical applications are limited to the range of Mach numbers where the shock wave is in the region where suction is applied.

Griffith aerofoils seem likely to have relatively high $C_{L\ max}$ at high Mach numbers and an investigation of the high speed characteristics of a Griffith aerofoil would be well worthwhile.

### 2. Sweptback wings

No experimental evidence on the $C_{L\ max}$ of sweptback wings at high M is available. Sweptback wings at high speeds however, show the same early tip stalling tendencies as they do at low speeds. Low speed wind tunnel tests with suction are to be made shortly and should give some indication of the practicability of this scheme at high M. The effect of sweepback is to increase the sectional $C_{L\ max}$ over the inner portion of the wing and a suitable remedy for the tip stalling may be sufficient to give high $C_{L\ max}$ at high M for sweptback wings.

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## 1 Introduction

Recent project work on some high speed high altitude designs has shown that one of the most important parameters deciding the size of an aircraft for a given manoeuvrability* is the $C_{L\,max}$ that can be attained at the high speeds considered. A knowledge of the maximum lift and the stalling incidence is also necessary when considering the effects of up gusts. For some wing sections $C_{L\,max}$ has been found to be seriously affected by compressibility; the available data have therefore been analysed in order to assist in the selection of wing sections with suitable characteristics.

The wings of some of the high speed designs now envisaged are sweptback and, from a stalling point of view, this is a complicating feature. This note is therefore divided into two main sections dealing with unswept and sweptback wings separately.

## 2 Unswept wings

### 2.1 Experimental evidence

Some flight measurements of the variation of $C_{L\,max}$ with Mach number are given in Figs.1 and 2 and all show large effects.

Fig.3 contains all the known wind tunnel data on the N.A.C.A.0012-63 section. Most of the experimental points are from three dimensional tests (Refs. 1, 2, 3 and 4) on aerofoils of aspect ratio 6 with taper ratio between 1.0 and 0.5. $C_{L\,max}$ first drops rapidly from 1.5 at $M = 0.2$ to about 0.8 at $M = 0.45$. This is followed by a further but more gentle drop to a value of 0.67 at $M = 0.8$, where $C_{L\,max}$ increases again. The lift carpet of this aerofoil (Ref.1) shows a peculiarity occurring fairly frequently in other similar high speed tests. The lift curves in the region $0.7 < M < 0.8$ exhibit subsidiary maxima somewhat lower than the true maxima. The reason for this is not clear as the corresponding pressure distributions are not available but it is thought to be caused by the relative displacement of the shock waves on both surfaces. Several aircraft in flight have been unable to reach their actual $C_{L\,max}$ either through loss of control power or through severe buffeting. Subsidiary maxima in the lift curves similar to those found on the N.A.C.A.0012-63 lift curves may be a possible cause of buffeting in flight and a practical limitation on $C_{L\,max}$.

Experimental points on the N.A.C.A.0012-63 section from independent sources are lacking above $M = 0.4$ but as no large differences in $C_{L\,max}$ would be expected between the two dimensional case and the tests with an aspect ratio of 6 the data from Ref.5 and 6 have also been plotted in Fig.3. These data indicate that the effect of Reynolds number is small above $M = 0.45$; below this value of $M$ the dotted curve of Fig.3 shows the probable variation of $C_{L\,max}$ with $M$ at constant Reynolds number.

Figures 1, 2, 3 and 4 contain all the known experimental data on the effect of Mach number on $C_{L\,max}$. The ad hoc nature of the tests makes it impossible to draw any general conclusions as to the effect of the various parameters. There is however a marked difference

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* Manoeuvrability is here used in its old sense i.e. that of ability to do manoeuvres. A measure of this is the maximum normal acceleration and therefore $C_L$ max that can be attained in level flight.

between the shape of the $C_L$ vs Mach number curves for conventional and low drag aerofoils as illustrated in Fig.2. At least up to $M = 0.7$ low drag aerofoils have higher $C_{L\,max}$ than conventional aerofoils of similar thickness-chord ratio. Also thin aerofoils show a smaller loss of $C_{L\,max}$ at high Mach numbers than do thicker ones.

## 2.2 Discussion

### 2.21 Low speed stalling properties of unswept wings

A qualitative insight into the stalling characteristics of a wing can be obtained by considering the growth of the boundary layer at high incidences (Ref.7). A typical potential flow pressure distribution at high incidences is shown in Fig.5a. The distance between A and B on the aerofoil surface is very small and it also shows a favourable pressure gradient so that along this part of the aerofoil the boundary layer will normally be laminar. Downstream of the point of maximum velocity the pressure gradient is unfavourable and depending on the Reynolds number and magnitude of the adverse pressure gradient, one of two things may occur, namely:

(a) The boundary layer will separate from the surface at B while remaining laminar. This is what usually occurs at low Reynolds number and results in a gentle stall because the flow does not permanently leave the surface but re-adheres as a turbulent layer further along. The maximum lift is poor and the lift incidence curve is flat topped.

(b) Transition will occur and because of its greater stability the turbulent layer will remain on the surface beyond B. Eventually the turbulent layer will begin to separate at the trailing edge and depending on the shape of the pressure distribution the separation will spread forward either gradually or abruptly giving a gentle or a sharp stall. In either case $C_{L\,max}$ will be considerably higher than if laminar separation occurred.

The effects of thickness chord ratio, camber and leading edge radius of curvature can also be understood by examining their respective effects on the pressure distribution. This has been done partially by Young in Ref.8, and the main conclusions are:

(a) Camber increases $C_{L\,max}$

(b) An increase in thickness above 9% also increases $C_{L\,max}$.

(c) Bringing the position of the maximum camber forward is detrimental.

(d) Small leading edge radius of curvature ($< 0.005c$) usually gives a relatively low but gentle type of stall provided that camber is small and the wing thickness is not too large.

(e) Leading edge radius of curvature above 0.01c usually gives higher maximum lift but the character of the stall is likely to be sudden.

### 2.22 Effects of compressibility on $C_{L\,max}$ of unswept wings

At velocities below the appearance of shock waves the main effect of compressibility is to increase the potential flow pressures in incompressible flow by $\frac{1}{\sqrt{1-M^2}}$ and therefore to increase suction peaks and pressure gradients in that ratio. As there is no evidence to suggest that the boundary layer is more stable in compressible than in incompressible flow the boundary layer will, if the transition point remains unaltered, separate under the same adverse pressure gradient and therefore at the same lift coefficient. No appreciable effect on $C_{L\ max}$ would therefore be expected except a slight reduction in the stalling incidence due to the increase in $\frac{dC_L}{d\alpha}$.

At higher values of M a region of supersonic velocities occurs on the upper surface bounded at the rear by a shock wave. Ackeret (Ref.9) has shown that the shape of this shock wave is determined by conditions in the boundary layer ahead of it; a forked shock wave forming with laminar layers and a straight shock wave forming with turbulent layers. He also shows that if the boundary layer is laminar ahead of the shock wave transition occurs between the oblique and the rear members of the shock wave so that the boundary layer behind a shock wave is always turbulent. We can therefore expect the effect of Reynolds number to be greatly reduced after the appearance of shock waves and this is supported by the experimental evidence on the N.A.C.A. 0012 section (Fig.3). The evidence on the Welkin and on the Spitfire of Fig.1 is not conclusive. On the Welkin the lift curves exhibit a kink at low Mach numbers and subsidiary maxima at high M so that the wind tunnel curves of Fig.1 may correspond to subsidiary maxima similar to those found on the N.A.C.A.0012 section. In the case of the Spitfire the true $C_{L\ max}$ was not attained because of severe buffeting.

Ackeret (Ref.9) also shows that the turbulent layer behind the shock wave is considerably thickened and therefore will tend to separate more easily. This results in a drop in $C_{L\ max}$ as has been found in all cases (Fig.1, 2, 3 and 4).

As M is further increased, the suction peaks are increased and the shock wave on the upper surface is displaced towards the trailing edge. A number of pressure plotting measurements in the R.A.E. and N.P.L. high speed tunnels (Ref.10) have shown that except for small local peaks there is a limiting value of the pressure ahead of the shock wave corresponding to about 0.3 of the total head of the undisturbed stream; an example of this is given in Fig.7. The pressure distributions of Fig.6 indicate that, except for the peaks, the maximum local M are of the order of 1.4 which corresponds to a $\frac{p}{H_o}$ of 0.31. An exception to this occurs at M = 0.68 on the N.A.C.A.23015 aerofoil and a possible explanation may be that the pressure distribution was taken just beyond the stall instead of at the stall. As the absolute pressure on the aerofoil is limited, the upper surface lift coefficient at a given Mach number depends mainly on the position of the shock wave. This is illustrated in Figs.5b and 6 where the $C_{L\ max}$ and the pressure distributions of a conventional and a low drag aerofoil are compared. The early rearward shift of the shock wave accounts for the higher $C_{L\ max}$ of the low drag aerofoil. A further comparison of $C_{L\ max}$ on conventional and low drag aerofoils is given in Fig.2.

Eventually the shock wave on the upper surface reaches the trailing edge and if, as Fig.6 indicates, the lift contribution of the lower surface is small, a constant $\frac{p}{H_o}$ of 0.3 on the upper surface gives a $C_{L\ max}$ of the order of 0.9 at M = 0.9.

2.3 Possible methods of improving $C_{L\,max}$

There appear to be two main effects of compressibility on $C_{L\,max}$.

(a) The first is purely a viscous effect occurring at relatively low Mach numbers due to the thickening of the boundary layer by the shock wave with consequent early separation. The only possible remedy for this is to apply suction at and immediately behind the shock wave. In practice this would only be possible for a small range of positions of the shock wave and therefore for a small range of Mach numbers. The best suction method in this case would probably be distributed suction over a portion of the upper surface through a porous material (Ref.11-12). Although of limited practical application the testing of this device would be of use and as most of the experimental evidence available is on the N.A.C.A.0012-63 section the suction could conveniently be tested on this aerofoil.

(b) The second effect is a limitation of the value of the pressure ahead of the shock wave which occurs at fairly high Mach numbers. $C_{L\,max}$ is then mostly a function of the position of the shock wave and the further back it is the higher $C_{L\,max}$ will be. The position of the maximum velocity at high $C_L$ should therefore be as far back as possible. Means of obtaining a far back position of the maximum velocity are: (a) to use a section having a far back position of the maximum thickness. (b) to use a thin cambered section. Adoption of (a) normally leads to large trailing edge angles with consequent undesirable pitching moment characteristics at high speeds (Ref.1) so that the furthest back position of the maximum thickness which can be used is probably only 40% of the chord from the leading edge.

A possible way of obtaining a far back position of the maximum thickness without large trailing edge angles is to have a discontinuity in the pressure distribution near the trailing edge as is done on the Griffith aerofoil. The high speed characteristics of Griffith aerofoils are not known and should be investigated. The use of camber introduces undesirable tail loads at high speeds and some sort of compromise would have to be made on the basis of systematic tests.

2.4 Conclusions

1. The experimental evidence available is scanty and its ad hoc nature makes it impossible to deduce the effect of the various parameters influencing the aerofoil shape. The only conclusions possible by examination of the data are:-

(a) At a given Mach number, the $C_{L\,max}$ of sections with far back position of the maximum thickness is higher than that of conventional sections owing to the further back position of the upper surface shock wave.

(b) At high Mach numbers thin aerofoils have higher $C_{L\,max}$ than thicker ones.

2. Systematic research on the influence of profile shape is needed and as the effect of Reynolds number appears small at high Mach number the systematic tests could conveniently be made in the N.P.L. high speed tunnel. The most important parameters on which evidence is required are thickness-chord ratio, position of maximum thickness and camber.

3. Distributed suction over the forward part of the wing upper surface may improve $C_{L\,max}$ over a small range of Mach numbers by removing the thickened boundary layer immediately behind the shock wave. Although of limited practical application the testing of this device would be of use and as most of the available experimental evidence is on the N.A.C.A.0012-63 section the tests could conveniently be made on this aerofoil.

4. In view of the far back position of the maximum velocity at high $C_L$ obtained on Griffith aerofoils the high speed characteristics of these sections should be investigated.

## 3 Sweptback wings

### 3.1 Experimental evidence

No measurements of $C_{L\,max}$ on sweptback wings at high Mach number are known. Two cases are known however when lifts near the $C_{L\,max}$ were measured: in Ref.21 a $C_L$ of 0.6 was measured at M = 0.84 on a 14% thick symmetrical wing section ( maximum t/c at 0.4c) with 45° of sweepback and in Ref.22 a $C_L$ of 0.64 was reached at M = 0.84 on a 12% symmetrical wing section (maximum t/c at 0.3c) with 45° of sweepback. In both cases the actual $C_{L\,max}$ was not reached but both lift curves showed some curvature.

### 3.2 Discussion

#### 3.21 Low speed stalling properties of sweptback wings

All the remarks made in para. 2.21 concerning the variation of the type of stall with Reynolds number apply also to the swept wing.

Theoretical calculations and tunnel measurements (Ref.23-24) have shown that, at a fixed incidence, sweepback results in a net loss of total lift with the greatest loss taking place at the centre of the span. At moderately large angles of sweepback this results in a spanwise distribution of local $C_L$'s showing a maximum usually in the wing tip region. Furthermore McKinnon Wood (Ref.25) and Griffith (Ref.26) have shown that sweepback has two main effects on the boundary layer, namely:

(a) A decrease in the effective pressure gradient due to the inclination of the flow to the isobars.

(b) A side travel of the boundary layer caused by the pressure gradient normal to the flow. On the inner part of the wing this side travel is outwards, near the tip the side travel is inwards thus causing a local thickening of the boundary layer.

The first effect is favourable from the point of view of the boundary layer stability; the second is unfavourable. The worst possible case therefore occurs when the isobars are normal to the main stream because then the adverse effect only is present and the boundary layer is thickened by inward flow from both sides. Such a condition usually occurs in the wing tip region. This, together with the increased local $C_L$ at the tips, is thought to account for the premature stalling of sweptback wings.

Fig.8 gives the results of a pressure plotting investigation on a sweptback wing in the R.A.E. 24 ft. tunnel at a Reynolds number of $1.55 \times 10^6$ (Ref.24). The wing plan form and sections are shown in the same figure. These results are not fully corrected and Fig.8 should only be taken qualitatively. Below the tip stalling incidence the spanwise distribution of local lift coefficient shows a maximum at about 0.7 of the semi-span from the centre line. After the tip stall the load is redistributed and the maximum occurs at 0.4 of the semi-span from the centre line - (Fig.9). $C_{L\ max}$ for the tip sections is low and the lift curve is flat topped whilst the local $C_{L\ max}$ at the centre sections is higher than would be measured in a two-dimensional test. This is in agreement with the earlier remarks on the effect of sweepback on the boundary layer because at the centre sections only the stabilising effect due to the reduction in the effective pressure gradient is present. The $C_{L\ max}$ of a sweptback wing is therefore a complicated function of the characteristics of the root and tip sections modified by the plan form and Reynolds number. A typical variation of the spanwise lift distribution with incidence on a sweptback wing is illustrated in Fig.9. Some curves of $C_{L\ max}$ against angle of sweepback (Ref. 21, 27, 28 and 29) are shown in Fig.10.

### 3.22 High speed stalling on sweptback wings

The general remarks of para.2.22 on the steepening of the pressure gradients by compressibility apply also in this case. Because of symmetry the relieving effects of sweepback are least at the centre section and, as has been observed in some German tests, a shock wave starts at the centre section and spreads outwards.

Figs.11 and 12 show pitching moment measurements made in the R.A.E. high speed tunnel on a symmetrical 14% thick wing section (max. t/c at 0.3c) aspect ratio of 5.8 and taper ratio of 0.57 at various angles of sweepback (Ref.21). The wing was tested with a fuselage. The low speed pitching moment curves at 35° and 45° sweepback indicate a tip stall in the neighbourhood of $C_L = 0.60$ and 0.50 respectively. At $M = 0.82$ the tip stall occurs at $C_L = 0.45$ and 0.5 respectively, indicating that part at least of the problem of the loss of stability at high M is the low speed problem of early tip stalling.

The fact that the final stall on a sweptback wing is preceded by a tip stall is of special importance in the case of tailless designs. Recent high speed tunnel tests on the DH.108 (Ref.30) indicate that although $C_{L\ max}$ is probably in excess of 0.6 at $M = 0.84$ the maximum $C_L$ that can be trimmed is only of the order of 0.45 owing to the loss of elevon power due to tip stalling.

## 3.3 Conclusions

Although no measurements of $C_{L\ max}$ on sweptback wings at high Mach number are known, the final stall of the wing is preceded by the stalling of the tip sections which limits the trimmed $C_{L\ max}$ of the wing particularly on tailless designs. This is the same problem already experienced at low speeds and a suitable solution should first be found there. Suction at the critical points for the boundary layer separation appears to be the most promising remedy and various arrangements of slots are to be tested at low speeds (Ref.31). These tests should give an indication of the quantities of air required and of the practicability of the scheme. Owing to the favourable effect of sweepback over the inner part of the wing the local $C_{L\ max}$ at high M is likely to be high at the inboard sections so that attention need only be paid to the tip sections. Still higher values of $C_{L\ max}$ could possibly be obtained by using suction in the critical region in conjunction with any of the previously mentioned methods to improve $C_{L\ max}$ of unswept wings.

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Attached:

Appendix
Table I
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# APPENDIX

## Aerofoil Notation

### A. N.A.C.A. Notation for Conventional Sections

1. Ref. N.A.C.A. Technical Report No.460.

x, y, z -- a,b.

- x Maximum camber (%)
- y Position of maximum camber (1/10 ths)
- z Maximum thickness (%) (2 Figs.)
- a Leading edge radius index
- b Position of maximum thickness (1/10 ths)

Values of a,
- 0 Sharp leading edge
- 3 1/4 normal
- 6 Normal $\left[\frac{\rho}{c} = 1.1\left(\frac{t}{c}\right)^{2}_{max}\right]$
- 9 3 times normal.

**Example:** 2409 -- 34 2% camber at 40% chord, 9% thick, max $t/c$ at 40% chord. 1/4 normal leading edge radius.

2. Ref. N.A.C.A. Technical Report No.537.

x, y, z.

- x maximum camber (%)
- y position of maximum camber × 2 (%) (2 Figs.)
- z maximum thickness (%) (2 Figs.)

**Example:** 23012 2% camber at 15% chord, 12% thick. These sections all have their maximum thickness at 30% chord.

### B. N.A.C.A. Notation for Low Drag Sections

Ref. A.R.C.5427, R.A.E. Technical Note No.Aero.1019

x,y - - z - - a,b.

- x denotes family to which section belongs
- y position of pressure minimum (1/10 ths)
- z amplitude of optimum $C_L$ range, measured from design $C_L$ (1/10 ths)
- a design $C_L$ (1/10 ths)
- b maximum thickness (%) (2 Figs.)

**Example:** 66 - - 2 - - 216 6 family, peak suction at 60% chord.

lift range (optimum) = 0.2 Design $C_L$ 0.2 - 16% thick
In the original notation z was omitted.

## C. British Notation for Low Drag Sections

1. x,y, / a, b

x maximum thickness (%) (2 Figs.)
y position of max. thickness "
a maximum camber (1/1000 ths) "
b position of maximum camber (%) "

**Example:** 1240/0640 12% thick at 40% chord, 0.6% camber at 40% chord.
These numbers may be preceded by combinations of the letters E, C, Q, H. E = elliptic referring to nose shape as far as position of maximum thickness

C = cubic )
Q = quartic ) referring to tail shape
H = hyperbolic)

2. Ref. R.A.E. Technical Note No.1308

For sections designed according to the approximate aerofoil theory with velocity distribution of the "roof-top" type. The section is defined as:

$2LX_1$; a, b, c; $2LX_1^1$; $a^1$ $b^1$ $c^1$; $C_L$

2L means that the velocity distribution designed for consists of 2 straight lines

$aV_0$ excess of velocity at the nose

$bV_0$ maximum velocity at the point $X_1$

$cV_0$ excess velocity at the trailing edge

2L means that the distribution of the pressure difference between upper and lower surface divided by $\frac{1}{2}\rho V_0^2$ consists of 2 straight lines.

$a^1$ $\dfrac{\text{pressure difference}}{\frac{1}{2}\rho V_0^2}$ at the leading edge

$b^1$ $\dfrac{\text{maximum pressure difference}}{\frac{1}{2}\rho V_0^2}$ at point $X_1^1$

$c^1$ $\dfrac{\text{pressure difference}}{\frac{1}{2}\rho V_0^2}$ at the trailing edge

$C_L$ value of the low speed lift coefficient at which the specified loading is obtained.

## TABLE I

Wing sections of American aircraft on which the measurements of $C_{L\ max}$ given in Fig.2 have been made.

| Aircraft | Root Section | Tip Section |
|---|---|---|
| **CONVENTIONAL SECTIONS** | | |
| P-38F (Lightning) | N.A.C.A.23016 | N.A.C.A.4412 |
| F6F-3 (Hellcat) | N.A.C.A.23015.6 (modified) | N.A.C.A.23009 |
| P-39N (Airacobra) | N.A.C.A.0015 | N.A.C.A.23009 |
| **LOW DRAG SECTIONS** | | |
| P-51B (Mustang) | N.A.C.A. North American Compromise 16% thick | N.A.C.A. North American Compromise 11% thick |
| P-63A (King Cobra) | N.A.C.A.66, 2X-116 a = 0.6 | N.A.C.A.66, 2X-216 a = 0.6 |
| YP-80A (Shooting Star) | N.A.C.A.65 - 213 | N.A.C.A.65 - 213 |

FIG. 1.

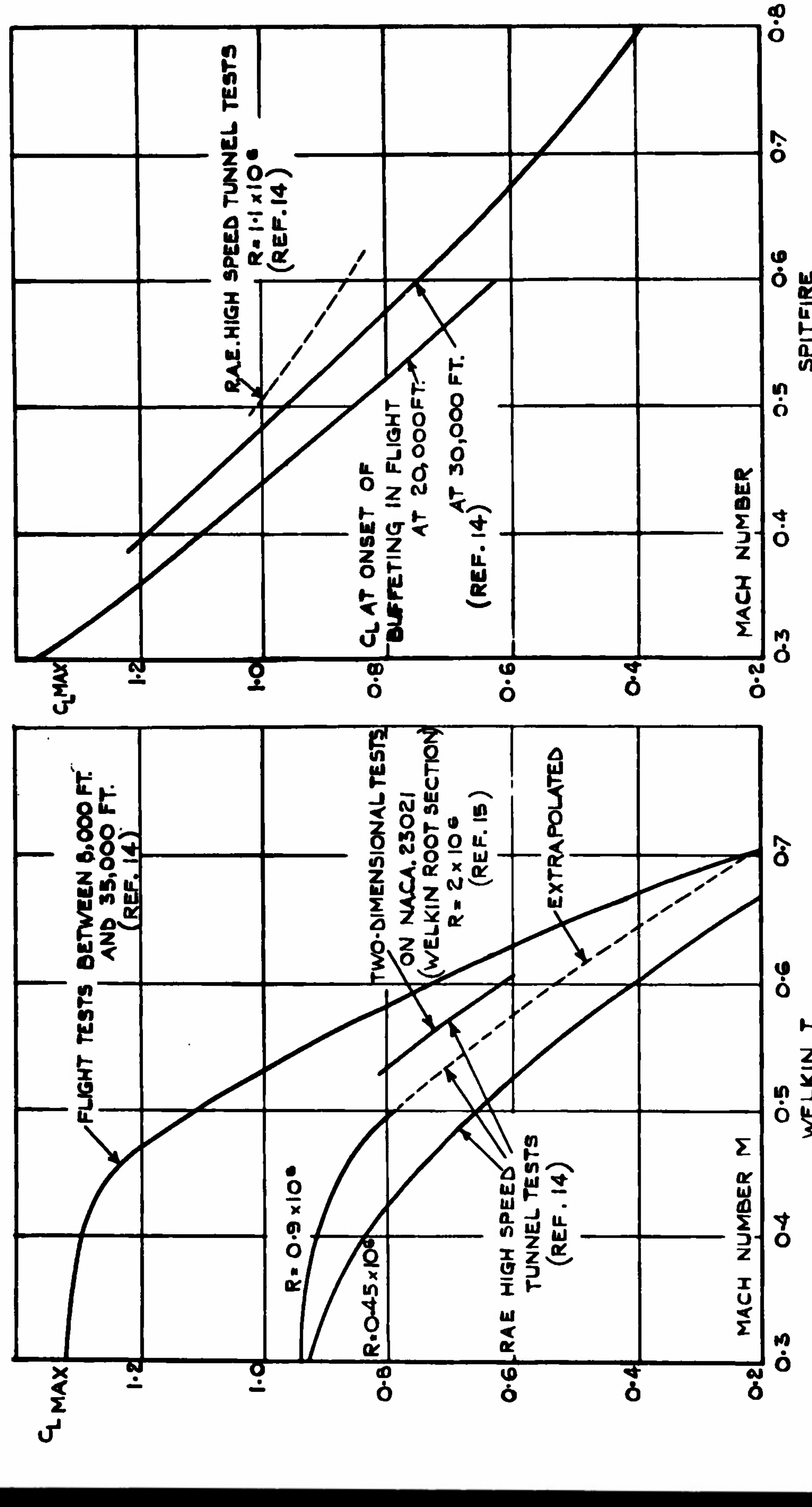


FIG. 1. EFFECT OF M ON $C_L$ MAX.

FIG. 2

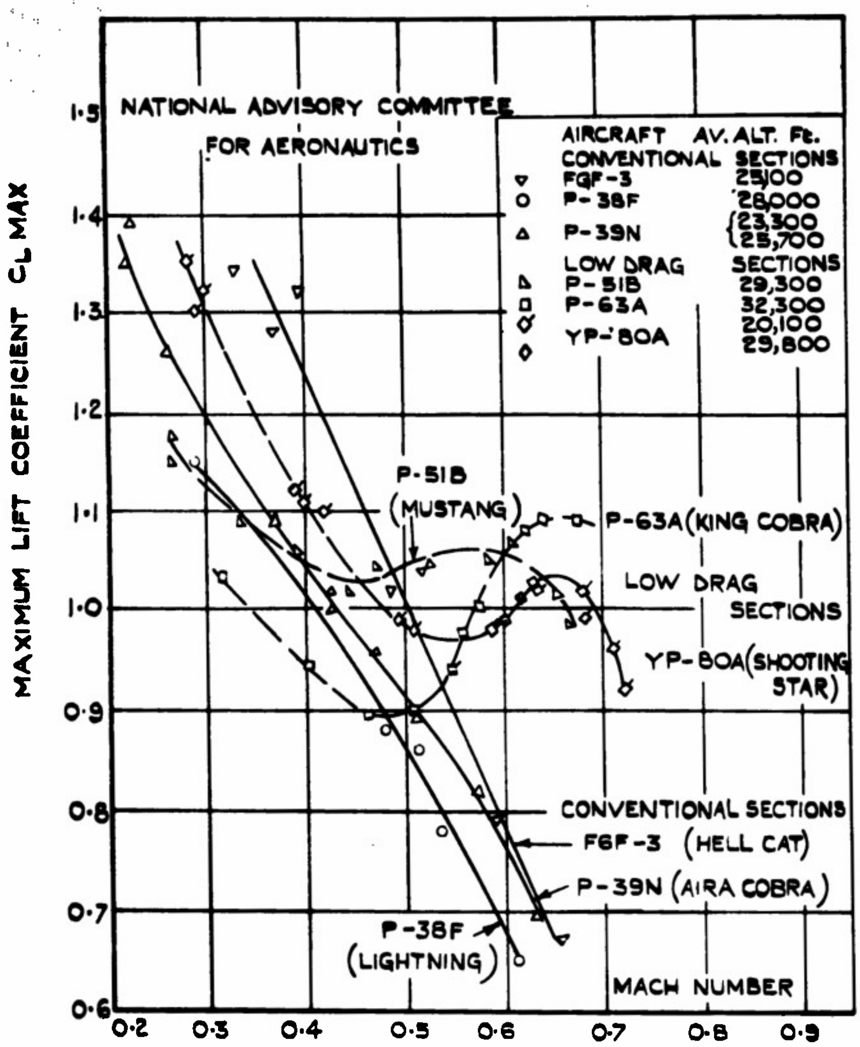


FIG. 2

VARIATION OE $C_L$ MAX. WITH MACH NUMBER

AMERICAN FLIGHT TESTS (REF. 16)

(WING SECTIONS IN TABLE I)
(NOTATION APPENDIX I)

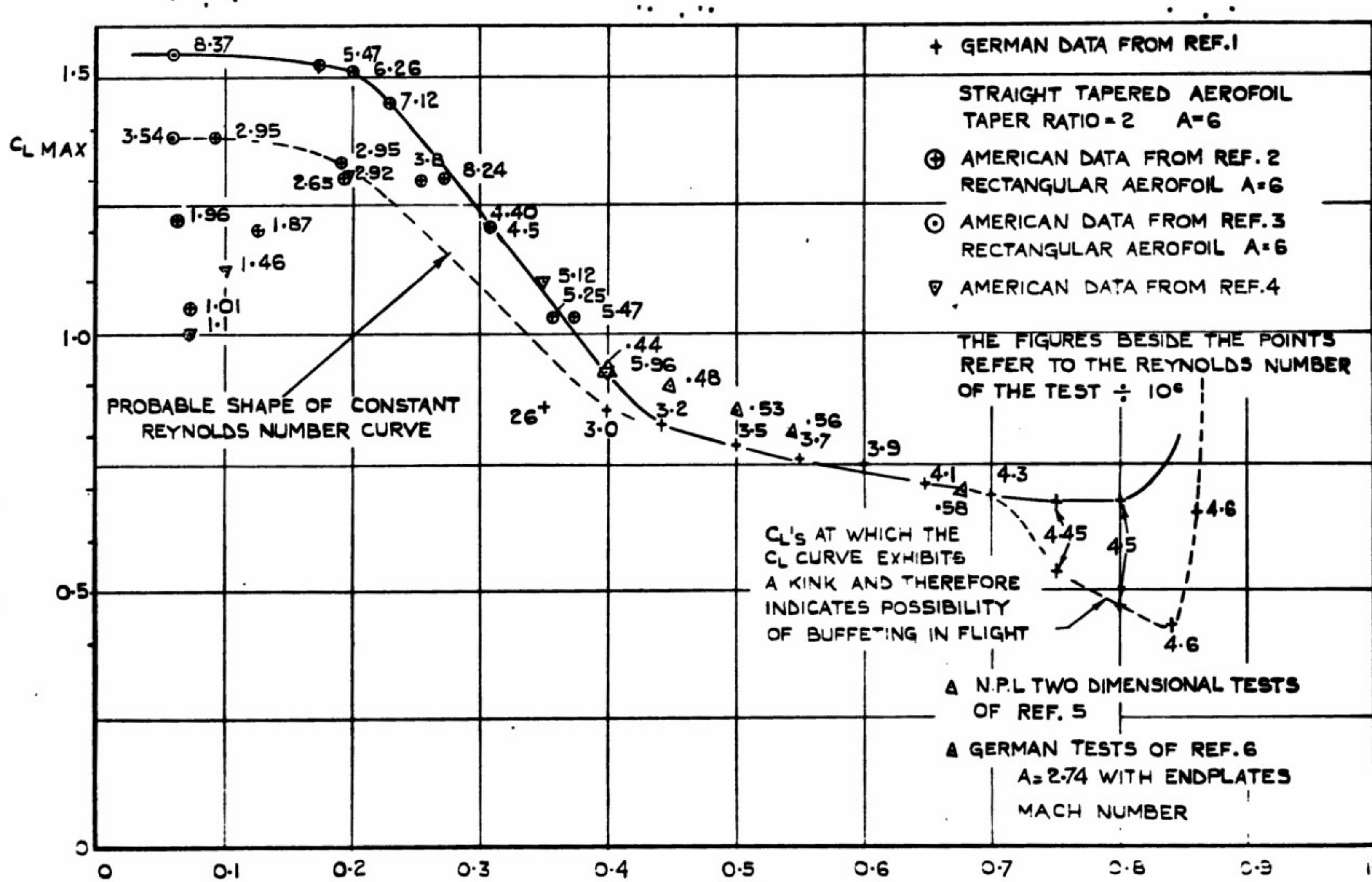


FIG.3. EFFECT OF MACH NUMBER ON $C_L$ MAX OF N.A.C.A.0012-63 AEROFOIL (MAX% AT 0·30c)

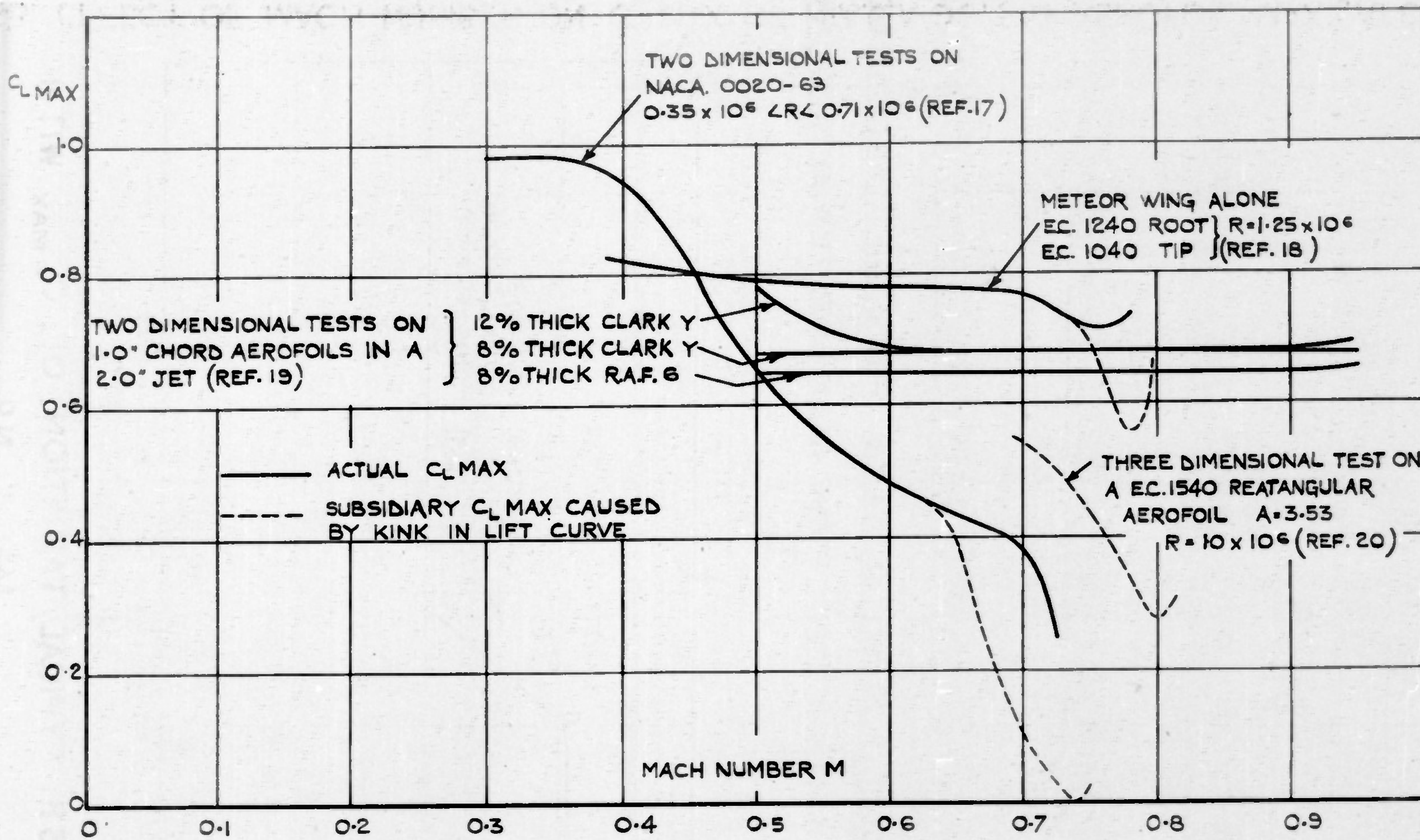


FIG. 4. EFFECT OF MACH NUMBER ON $C_L$ MAX.

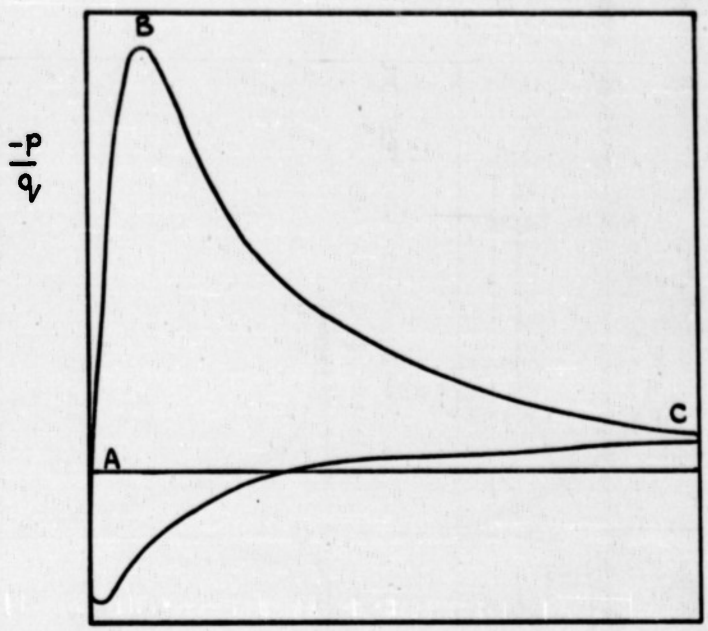


FIG. 5 A. POTENTIAL FLOW PRESSURE DISTRIBUTION AT HIGH INCIDENCE ON A CONVENTIONAL AEROFOIL

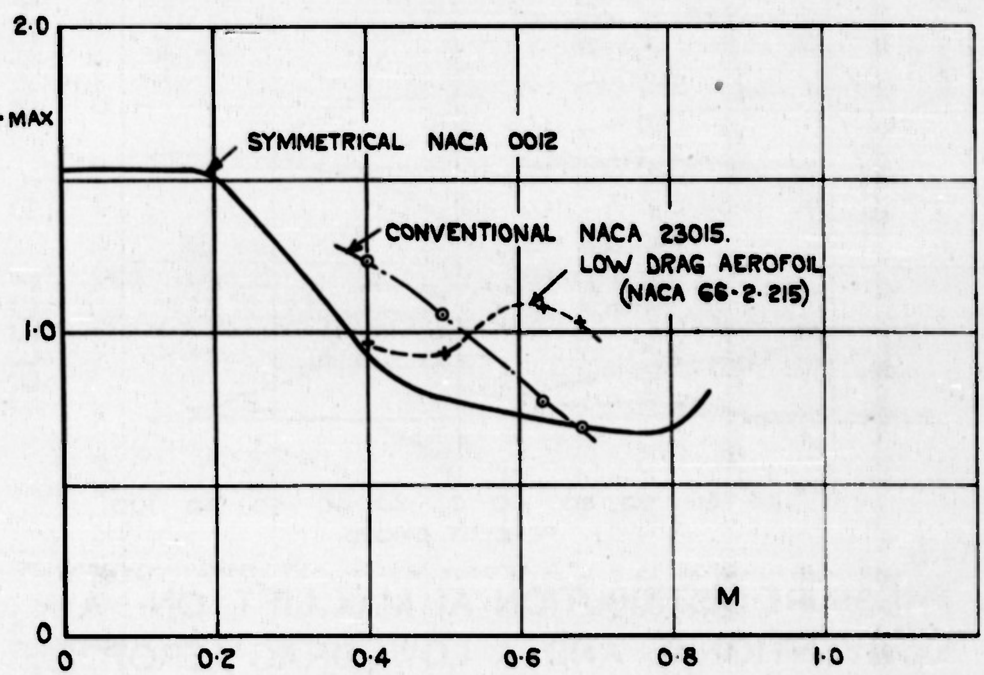


FIG. 5 B. TYPICAL VARIATION OF $C_{L\,MAX}$ WITH MACH Nº

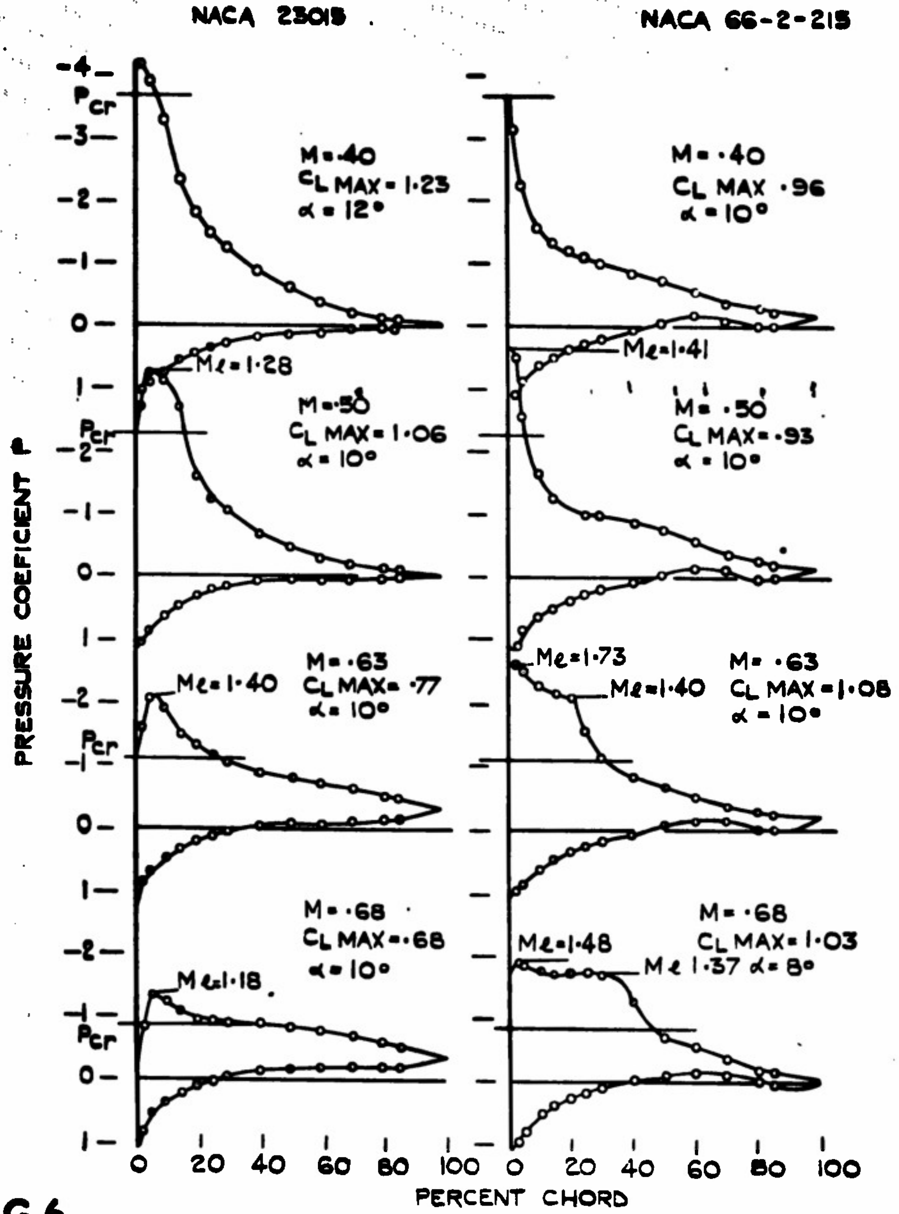


FIG.6

**PRESSURE DISTRIBUTION AT MAX.LIFT ON A CONVENTIONAL AND A LOW DRAG AEROFOIL**

(FROM REF. 16)

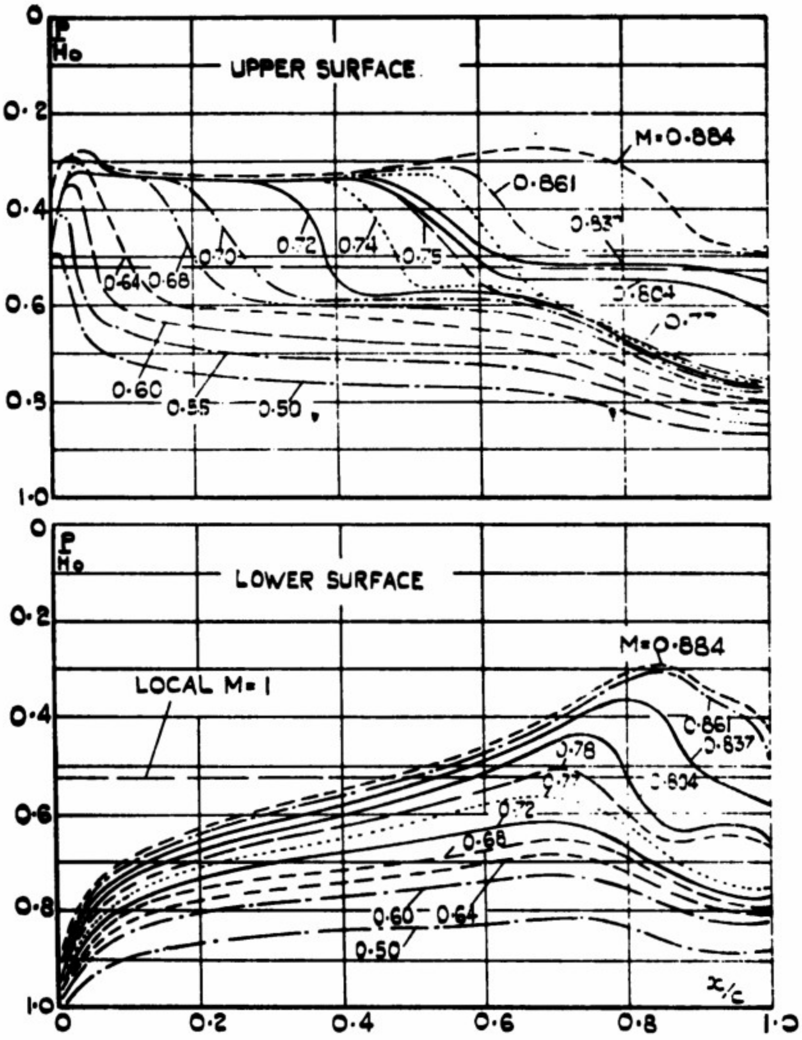


FIG. 7
PRESSURE DISTRIBUTION ON UPPER & LOWER SURFACES AT SEVERAL MACH NUMBERS
AEROFOIL N.A.C.A. 0 00 12 - 0·55 50/0·5 α = 6
(REPRODUCED FROM REF. 1)

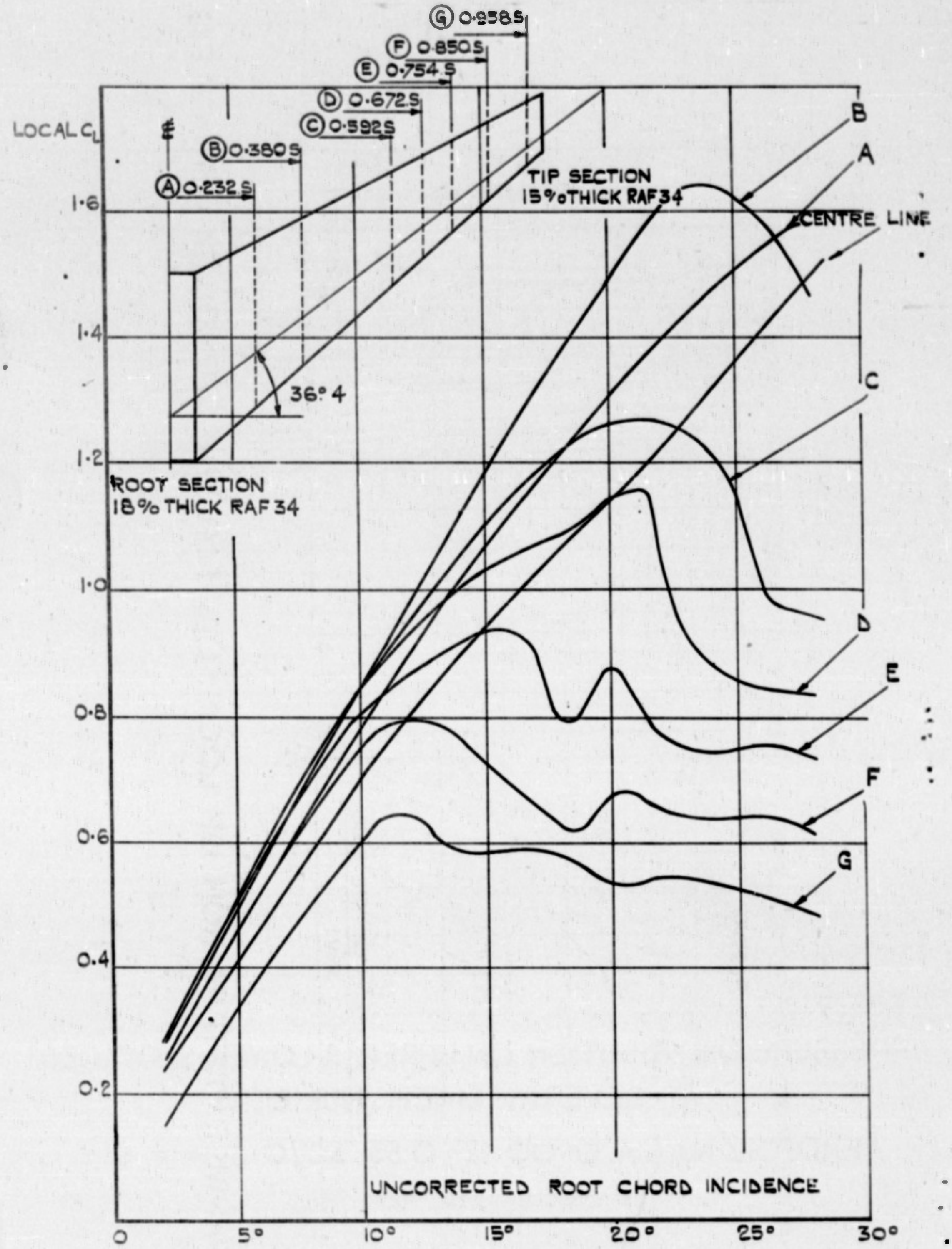


FIG 8. SPANWISE LOCAL LIFT CURVES ON A 36° SWEEPBACK WING. (REF. 24.)

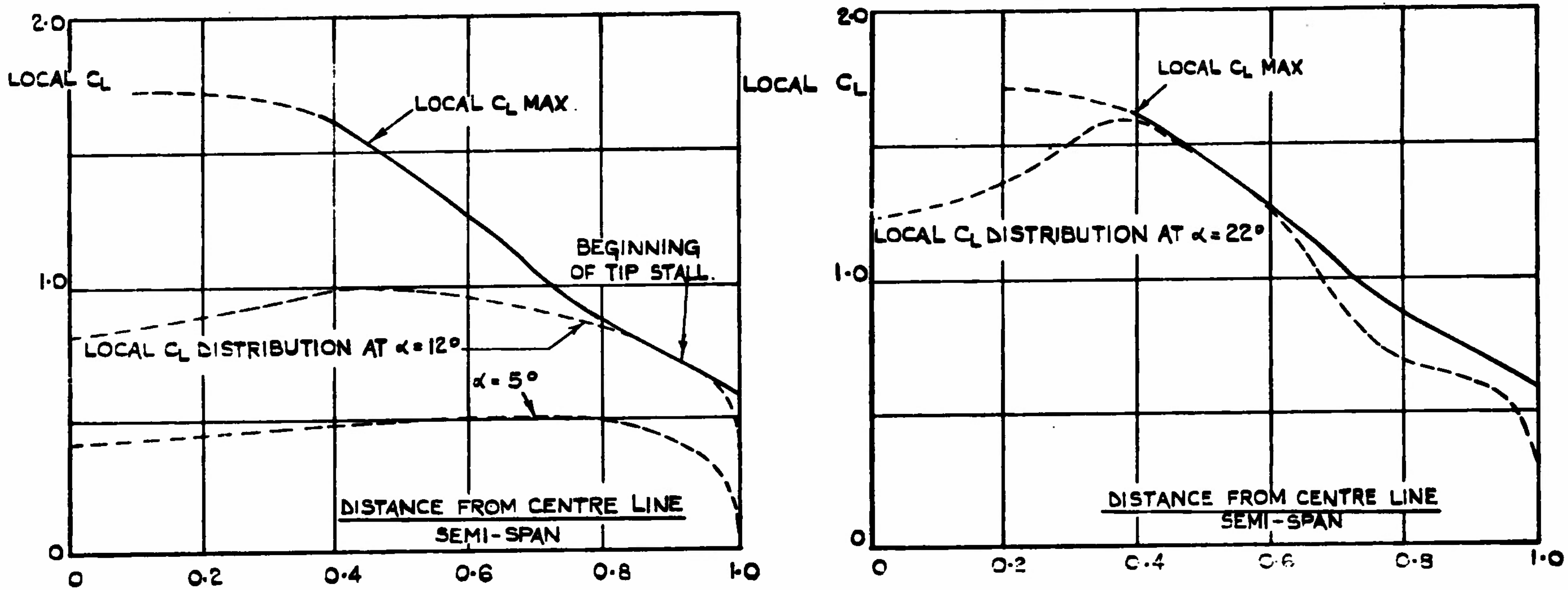


FIG.9. SPANWISE DISTRIBUTION OF LOCAL $C_L$'s FOR THE SWEPTBACK WING OF FIG.8

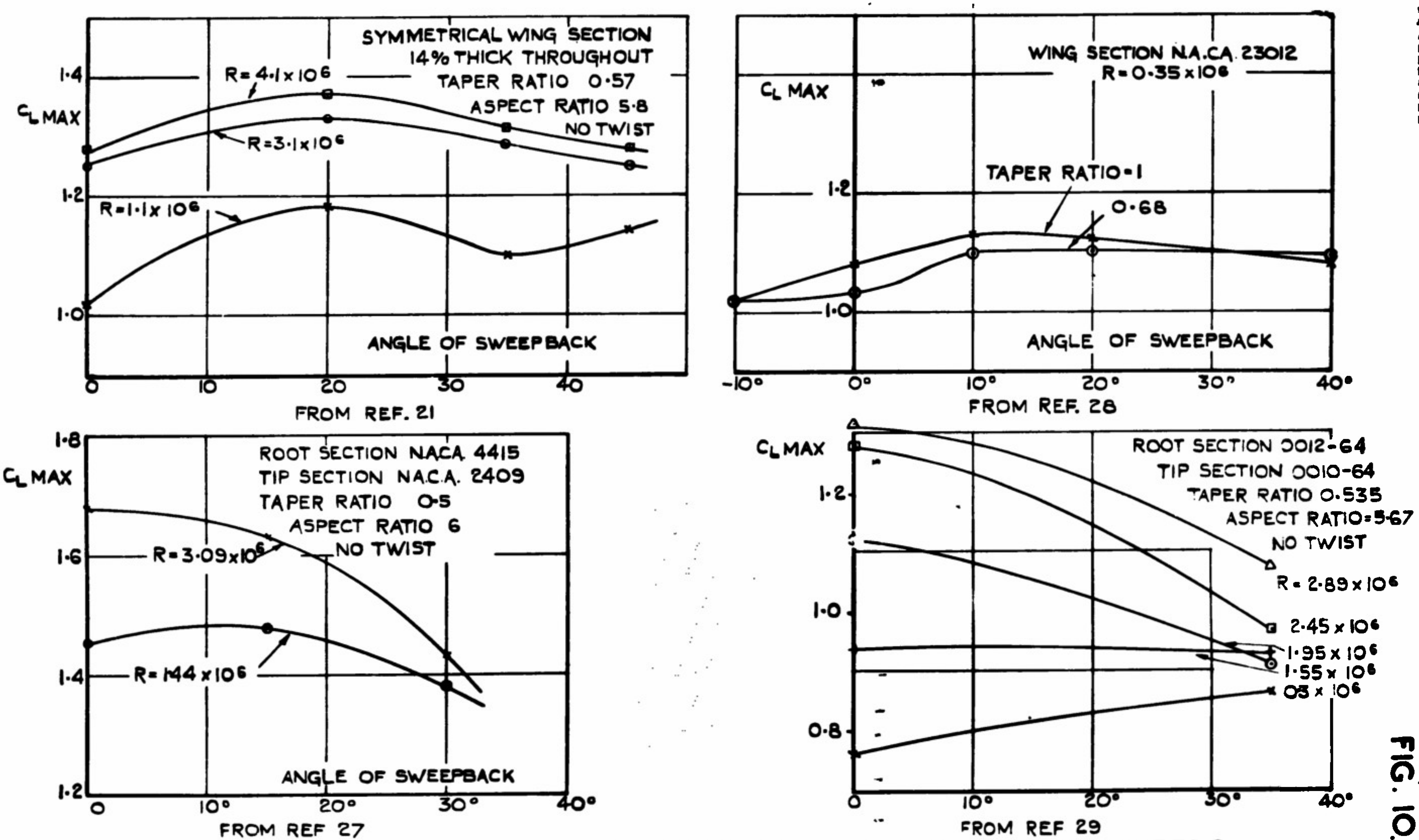


FIG. 10. SOME VARIATIONS OF $C_L$ MAX. WITH SWEEPBACK AT LOW SPEEDS.

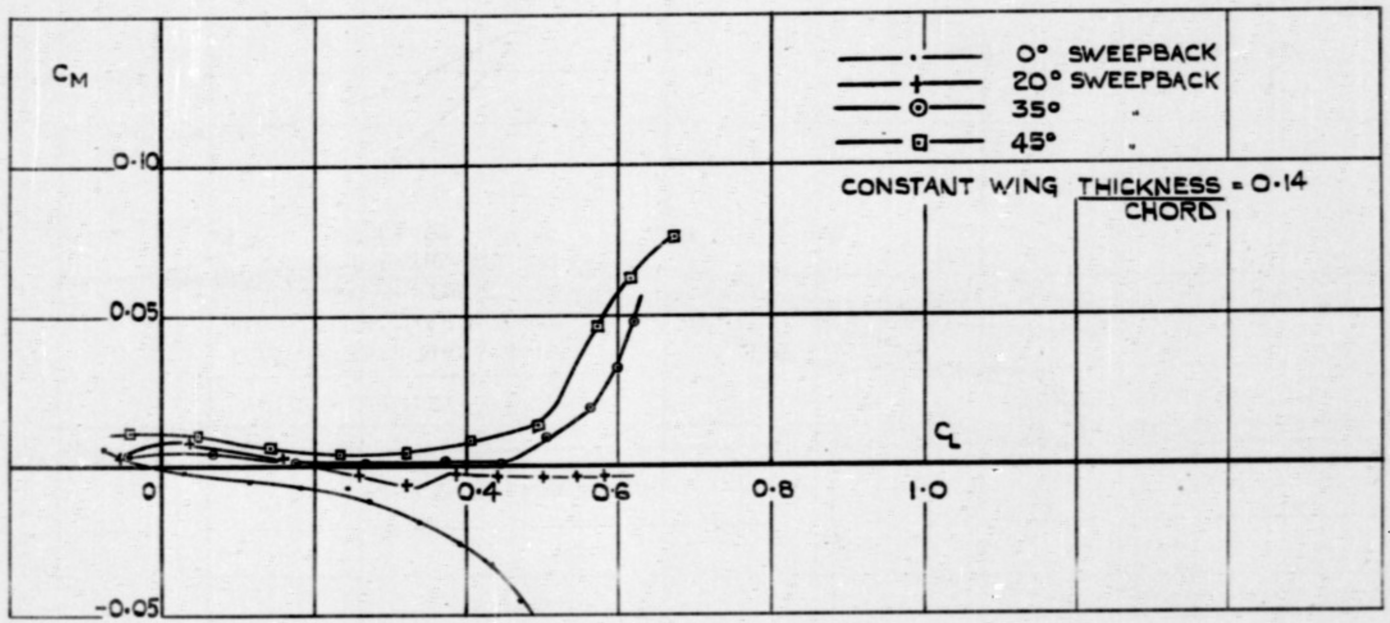


FIG.11. PITCHING MOMENTS AGAINST $C_L$ FOR WINGS OF VARIOUS SWEEPBACK (REF.21)

$M = 0{\cdot}8 \quad R = 1{\cdot}2 \times 10^6$

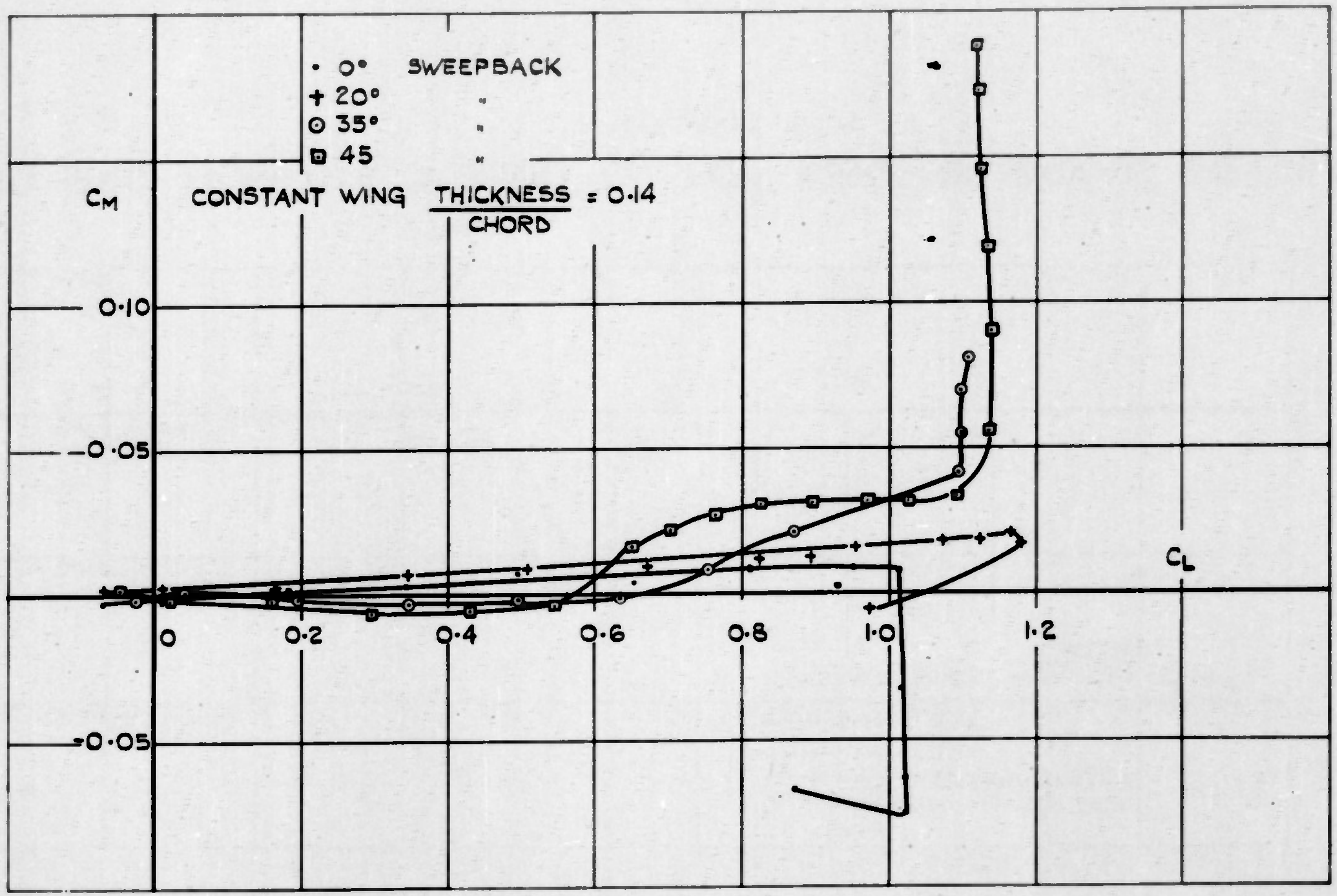


FIG. 12. PITCHING MOMENTS AGAINST $C_L$ FOR WINGS OF VARIOUS SWEEPBACK (REF 21)

$M < 0{\cdot}2 \quad R = 1{\cdot}2 \times 10^6$

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